6489
p-34

NASA Technical Memorandum 110170

Analysis of Selected Compression Splice Joint Locations in a Graphite-Epoxy Transport Wing Stub Box

D. C. Jegley
Langley Research Center, Hampton, Virginia

October 1995

National Aeronautics and
Space Administration
Langley Research Center
Hampton, Virginia 23681-0001

ANALYSIS OF SELECTED COMPRESSION SPLICE JOINT LOCATIONS IN A GRAPHITE-EPOXY TRANSPORT WING STUB BOX

Abstract

Three critical compression splice joint locations in a stitched graphite-epoxy transport wing stub box have been analyzed to determine their expected structural performance. The wing box is representative of a section of a commercial transport wing box and was designed and constructed by McDonnell Douglas Aerospace Company as part of the NASA Advanced Composites Technology (ACT) program. The results of the finite element analyses of the splice joints are presented. The analysis results indicate that failure will not occur in the splice joint regions for loads less than the Design Ultimate Load of the wing box.

Introduction

To evaluate the potential of a stitched graphite-epoxy material form for use on commercial transport aircraft wings, a short section of a wing box was designed and fabricated by the McDonnell Douglas Aerospace Company under the NASA Advanced Composites Technology Program (ACT) contract NAS1-18862. This short wing-box section is referred to herein as the "wing stub box." Current plans are to test the wing stub box at the NASA Langley Research Center to evaluate its structural performance. Several structural analyses were conducted by McDonnell Douglas Aerospace Company and by NASA Langley Research Center in support of this effort. The present paper describes the results of the analysis of three compression-loaded splice locations in the wing-stub-box test specimen.

Wing-Stub-Box Test-Specimen Description

The wing-stub-box structure consists of a metallic load-transition structure at the wing root, a composite wing stub box, and a metallic extension structure at the wing tip as shown in figure 1. The load-transition structure and the wing-tip extension structure are metallic end fixtures required for appropriate load introduction into the composite wing stub box during the test. The load-transition structure is located inboard of the composite wing stub box (between the composite wing stub box and the vertical reaction structure at the wing-stub-box root) and the wing-tip extension structure is located outboard of the composite wing stub box. The load-transition structure is mounted to a steel and concrete vertical reaction structure resulting in a near-clamped end condition. The entire structure, including the composite wing stub box and the metallic structures, is approximately 25 feet long. Details of the geometry of the structure are presented in reference 1.

The composite wing stub box was fabricated from Hercules, Inc. AS4/3501-6 and IM7/3501-6 graphite-epoxy materials which are stitched together using E. I. DuPont de Nemours Inc. Kevlar threads. IM7 graphite fibers are only used for the 0 degree fibers in the lower skin. The composite skin and stiffeners are composed of layers of the graphite material forms prekitted in nine-ply stacks that have a $[45/-45/0_2/90/0_2/-45/45]_T$ stacking sequence. Each nine-ply stack is approximately .058 inches thick. Several nine-ply stacks of the prekitted material are used to build up the desired thickness at each location. The composite wing stub box was fabricated using a Resin Film Infusion (RFI) process (see ref. 2).

Splice Description

Metallic splices are used to join the composite wing stub box to the metallic wing-tip extension structure and to the metallic load-transition structure (as indicated in figure 1). These metallic splices are bolted to the ends of the graphite-epoxy stiffeners and to the metallic extension and load-transition structures. Three splice locations on the upper cover panel of the composite wing stub box have been analyzed in detail and the results of the analysis of the three splices are presented in the present paper. The locations of the three splices are shown in figure 2. Splice 1 is located at the outboard end of the composite wing stub box. Splices 2 and 3 are located at the inboard end of the composite wing stub box.

Each splice consists of aluminum inner and outer splice plates and aluminum angle splices, as shown in figure 3. Bolts are used to attach the stiffener to the angle splices and the skin to the inner and outer splice plates. All stiffeners are approximately 2.3 inches high and .46 inches thick with a 1.12-inch-wide flange on either side of the blade. The distance between the end of the composite upper-cover-panel skin and the end of the composite stringer is 8.59 inches (as shown in figure 2) for all splices considered. However, the distance between the end of the composite stringer and the nearest rib is different for the three splices considered. The distances for splice locations 1, 2, and 3 are shown in figure 2 and are 6.1 inches, 6.4 inches and 2.8 inches, respectively.

Model Description

Three similar models were constructed to model accurately the geometry of the three splice locations considered. In each case, the entire 8.59-inch-long region of unstiffened skin and splice plates and 11 to 12 inches of the stringer is included in the model. The stringer is centered laterally for each model. The entire flange and 3 inches of skin is included on each side of the stringer, resulting in a region to be analyzed for each splice location that is approximately 20 inches long and 6 inches wide. The intercostal in the region of each splice is also included in the model.

The skin thickness at splice location 1 is nine of the nine-ply stacks of prekitted material. The skin thickness at splice locations 2 and 3 varies from eight to ten of the nine-ply stacks. The variation in skin thickness is included in the model. Quadrilateral and triangular finite elements are used to model the skin, blade and splice parts. Beam elements are used to model the bolts which connect the metallic parts to the composite parts. Each model has approximately 2,200 elements and 17,000 degrees of freedom. The models for the three splice locations are shown in figure 4.

The material properties for the stitched nine-ply stacks of the prekitted material used for the skin and blades are E_x=8.17 Msi, E_y=4.46 Msi, G_{xy}=2.35 Msi and ν_{xy}=.459 (ref. 3). The material properties used for the aluminum splice plates and angles are E=10.6 Msi, G=3.8 Msi and ν_{xy}=.395.

Analysis

Boundary conditions for the local compression-splice analyses were obtained from the global nonlinear finite element analysis results described in reference 4. Nodal displacements for each skin node at the local-splice-model edge were calculated from the global analysis displacement results. A nonlinear analysis of the splice regions was conducted using the STAGS finite element code (ref. 5). Maximum loads applied to the splice models correspond to the Design Ultimate Load (DUL) condition for the wing stub box.

Results and Discussion

Calculated displacements in the splices at splice locations 1, 2 and 3 are shown in figures 5, 6 and 7, respectively, for the DUL condition. Displacements shown in figures 5a, 6a and 7a are not magnified but are to the same scale as the model. Displacements in the axial (spanwise), lateral (chordwise), and out-of-plane directions are also shown in figures 5-7.

The outboard end of the cover panel skin for splice location 1 deforms out-of-plane 3.2 inches at DUL and the inboard end deforms out-of-plane 2.2 inches. Most of this displacement is due to the global bending of the wing stub box rather than local displacements in the joint region. The outboard end of the upper-cover-panel skin in splice location 2 deforms out-of-plane .16 inches at DUL while the inboard end deforms out-of-plane .11 inches. The outboard end of the cover panel skin in splice location 3 deforms out-of-plane .18 inches at DUL while the inboard end deforms out-of-plane .07 inches. The out-of-plane displacement at splice location 1 is much larger than the out-of-plane displacement at splice locations 2 and 3 since splice location 1 is farther away from the root of the wing stub box. The inplane displacements are much smaller than the out-of-plane displacements at splice location 1. The inplane displacements are of the same order of magnitude as the out-of-plane displacements at splice locations 2 and 3.

Calculated strains in the splices at splice locations 1, 2 and 3 are shown in figures 8, 9 and 10, respectively, for the DUL condition. The axial strains in the splices are shown in figures 8a, 9a and 10a, the lateral strains are shown in figures 8b, 9b, and 10b, and the shear strains are shown in figures 8c, 9c, and 10c. Away from the boundaries of the model, the maximum axial surface strains at DUL are approximately .008 in/in. The unnotched allowable strain for this material system given in reference 3 is .0093 in/in.

Away from the boundaries of the model, the maximum lateral strains at DUL are approximately .008 in/in. The maximum shear strains are .012, .004 and .008 in/in. for splice locations 1, 2 and 3, respectively when the structure is subjected to DUL.

Strain gages are to be attached to the wing stub box at the locations shown in figure 11 for the three compression splices considered and the strain gages are designated as strain gages A, B, C, D and E. Gages A and D are back-to-back gages on the composite upper-cover-panel skin. Gage B is located on the aluminum splice. Gage C is located on the aluminum inner splice plate and gage E is located on the aluminum outer splice plate. Analytical predictions of the strain at these strain gage locations are shown in figures 12, 13 and 14 for splice locations 1, 2 and 3, respectively. The load for the ordinates in the figures is the load applied at the outboard end of the wing-tip extension structure. The applied load corresponding to Design Ultimate Load is 166,000 lb. The load-versus-strain curves shown in figures 12-14 indicate that the maximum strain is predicted to occur at the location of strain gage A, and is .0081, .0082, and .0071 in/in. at DUL for locations 1, 2, and 3, respectively. The strains increase almost linearly with load, indicating that geometric effects do not cause nonlinear response. Applied loads of approximately 195,000 lb, 185,000 lb, and 220,000 lb would be required for the strain at strain gage A to equal the unnotched allowable strain of .0093 in/in. at splice locations 1, 2, and 3, respectively. These load levels correspond to margins of safety of .17, .11 and .32, which indicate that failure should not occur at any of these regions unless the load is 11 percent greater than Design Ultimate Load.

Since the margins of safety in the regions of the upper cover panel away from these splice locations, such as near the access door, are less than .05 (ref. 6), failure is not expected to occur in these splice regions

Concluding Remarks

Three compression splice joint regions of the upper cover panel of a graphite-epoxy wing stub box were analyzed using a finite element analysis. Loads up to Design Ultimate Load (DUL) were considered. The predicted maximum strain in each compression splice joint region at DUL is approximately .008 in/in., which is less than the allowable strain of .0093 in/in. for the material. The wing stub box is not expected to fail at these compression splice joint regions for applied loads up to 11 percent greater than DUL.

References

1. Markus, Alan M.; and Grossheim, Brian G.: Manufacturing Development for Stitched/RFI Transport Wing Stub Box Structures. Proceedings of the Fifth NASA/DoD Advanced Composites Technology Conference, NASA CP 3294, pp. 773-786, May 1995.

2. Markus, Alan; Thrash, Patrick; and Grossheim, Brian: Manufacturing Development and Requirements for Stitched/RTM Wing Structure. Proceedings of the Fourth NASA/DoD Advanced Composites Technology Conference, NASA CP 3229, pp. 503-523, 1993.

3. Hinrichs, S. C.: ICAPS Stub Box Structural Analysis Volume 1: Material Properties. McDonnell Douglas Aerospace Transport Aircraft-West, Report Number MDC94K9101, March, 1995.

4. Wang, John: Global and Local Stress Analyses of McDonnell Douglas Stitched/RFI Composite Wing Stub Box. NASA TM 110171, October 1995.

5. Brogan, F. A.; Rankin, C. C.; and Cabiness, H. D.: STAGS User Manual. Lockheed Palo Alto Research Laboratory, LMSC Report P032594, 1994.

6. Hinrichs, S. C.: ICAPS Stub Box Structural Analysis Volume 4: Upper Skin Strength Analysis. McDonnell Douglas Aerospace Transport Aircraft-West Report Number MDC94K9101, March, 1995.

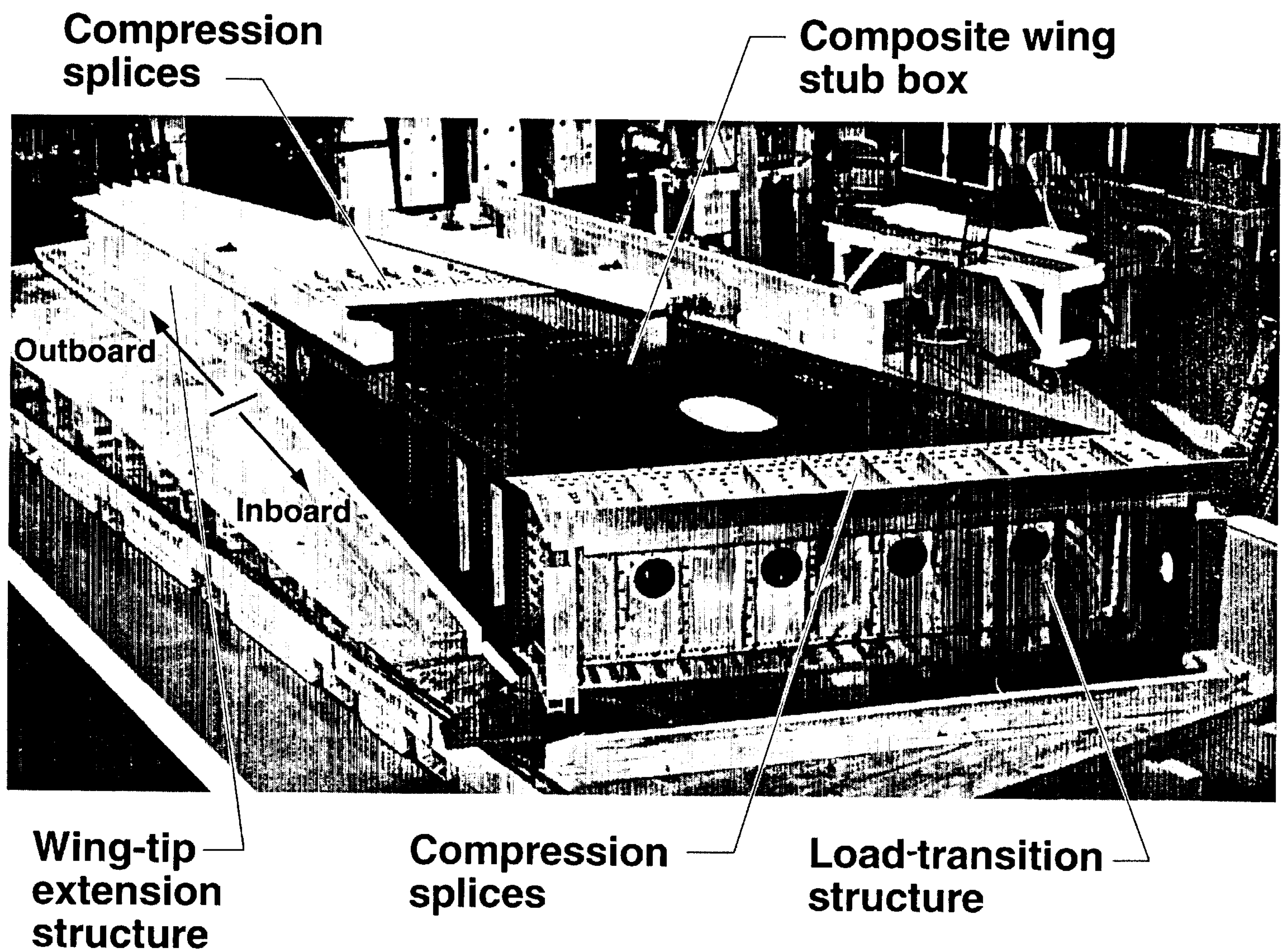


Figure 1. Composite wing-stub-box test-specimen.

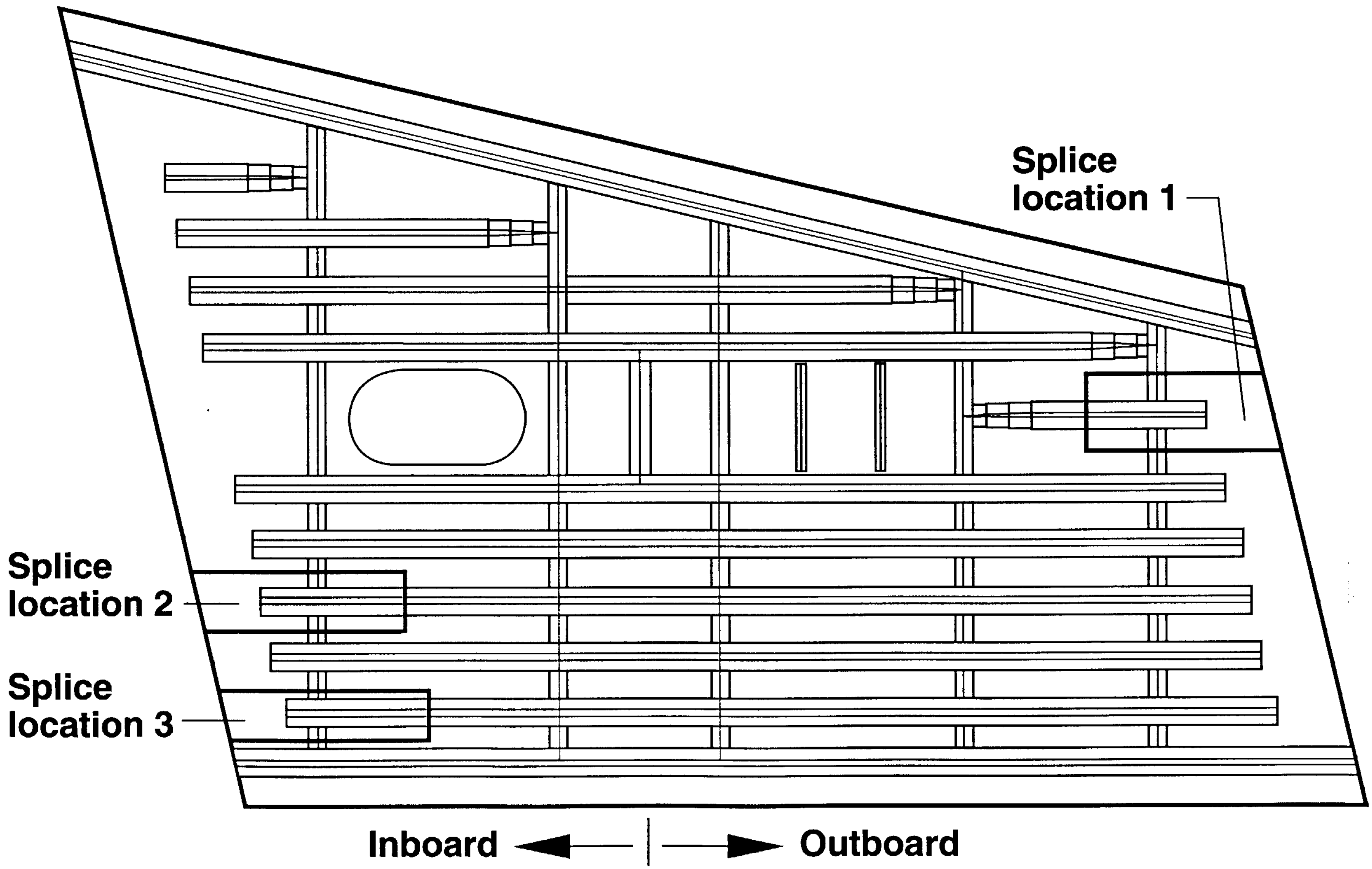


Figure 2. Compression splice locations.

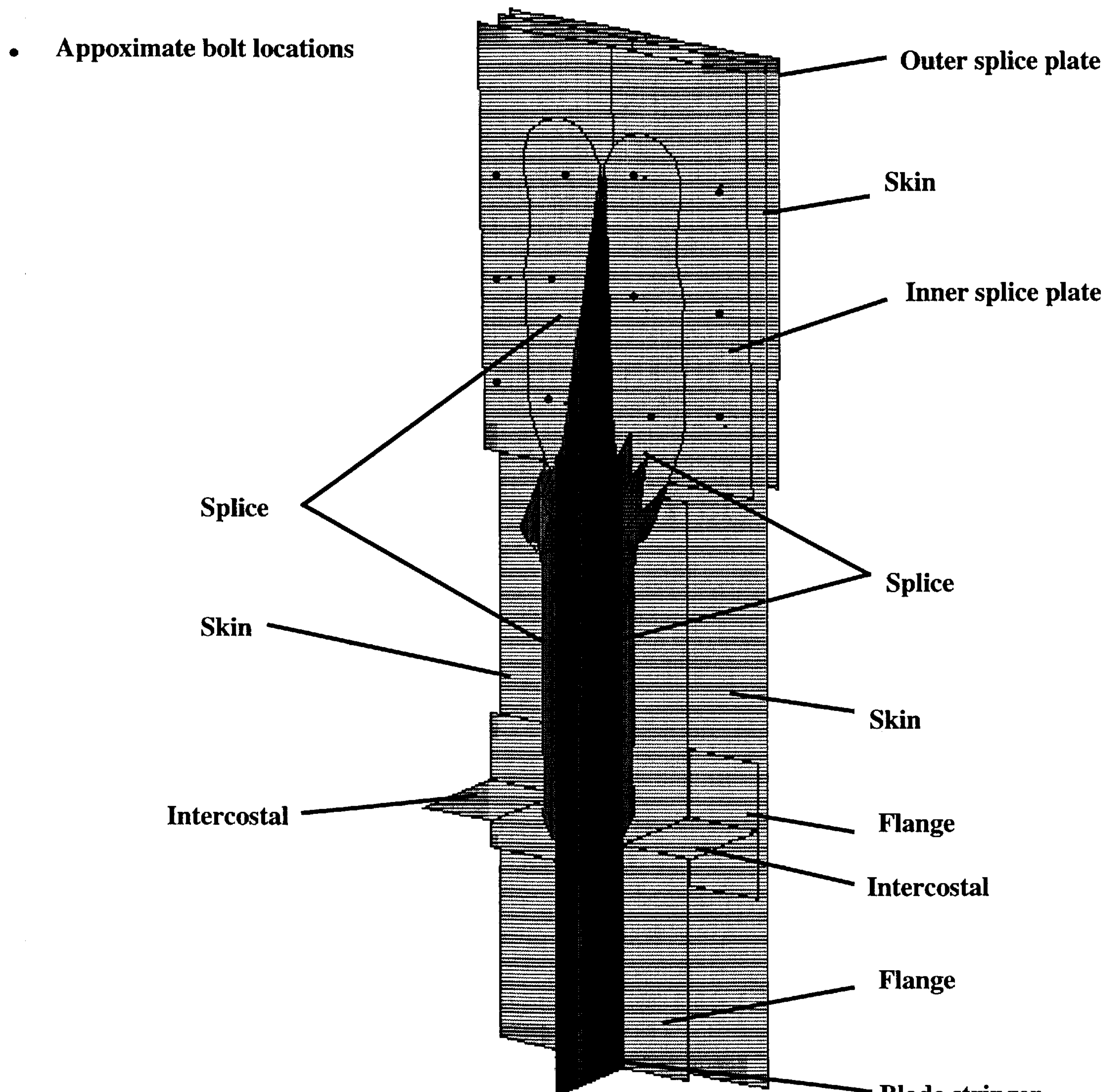
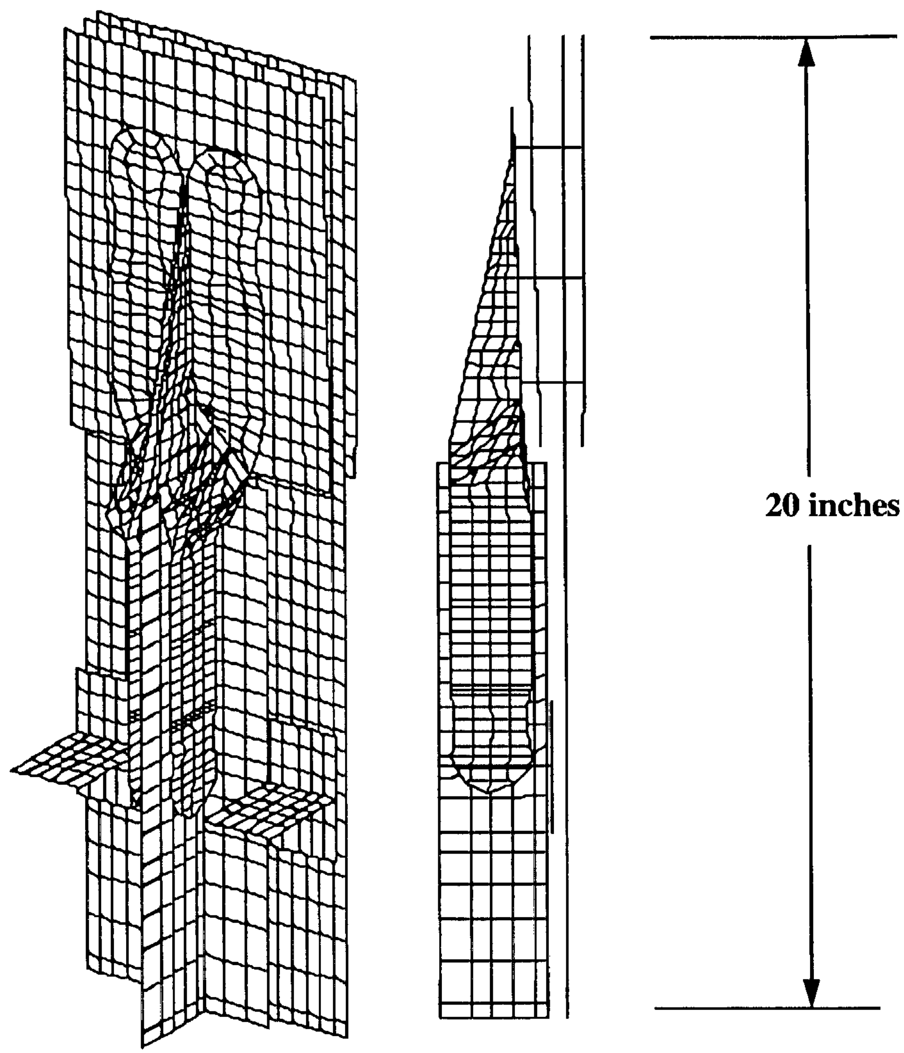
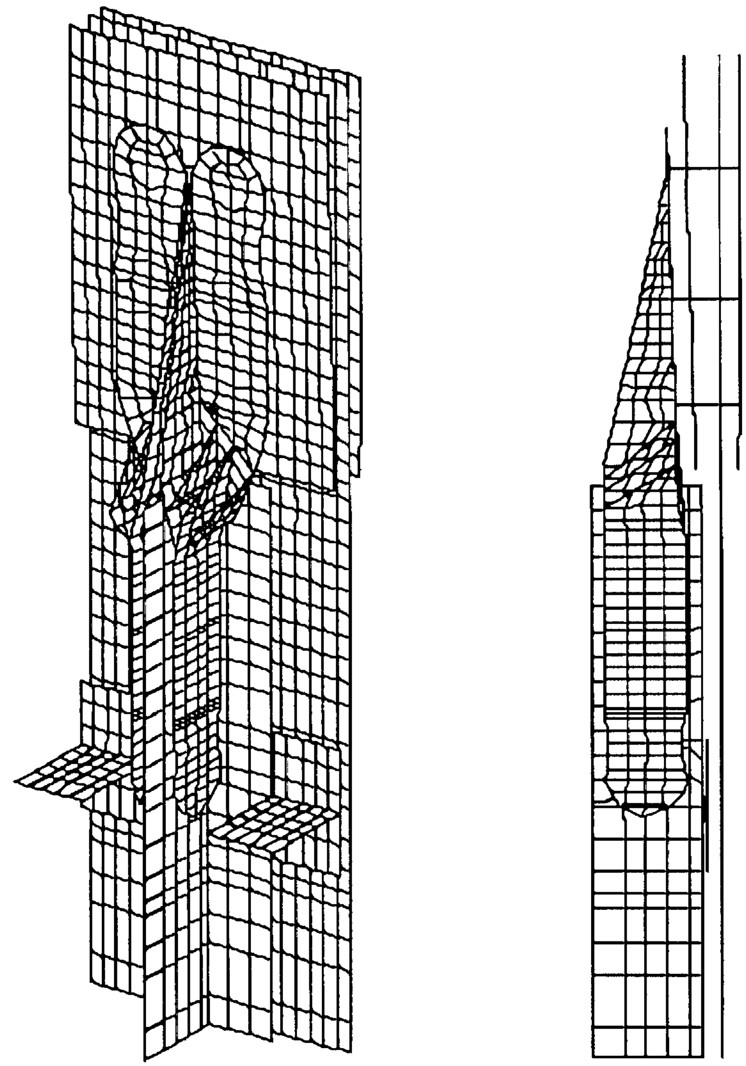


Figure 3. Splice details.

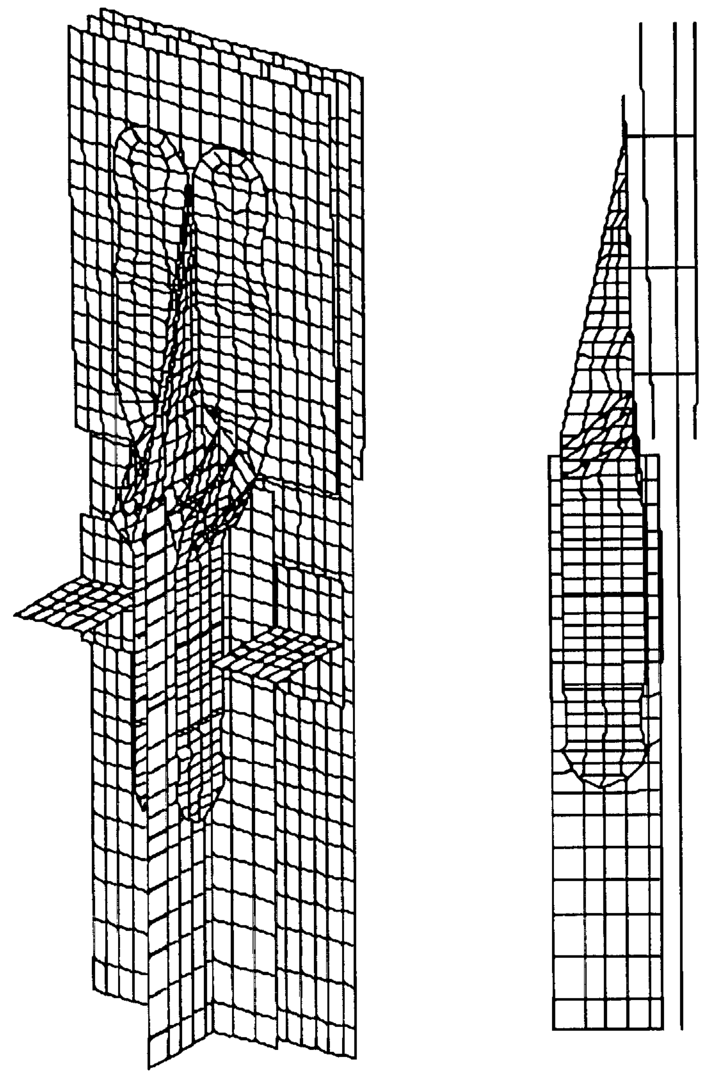


(a) Splice location 1.
Figure 4. Compression splice models.



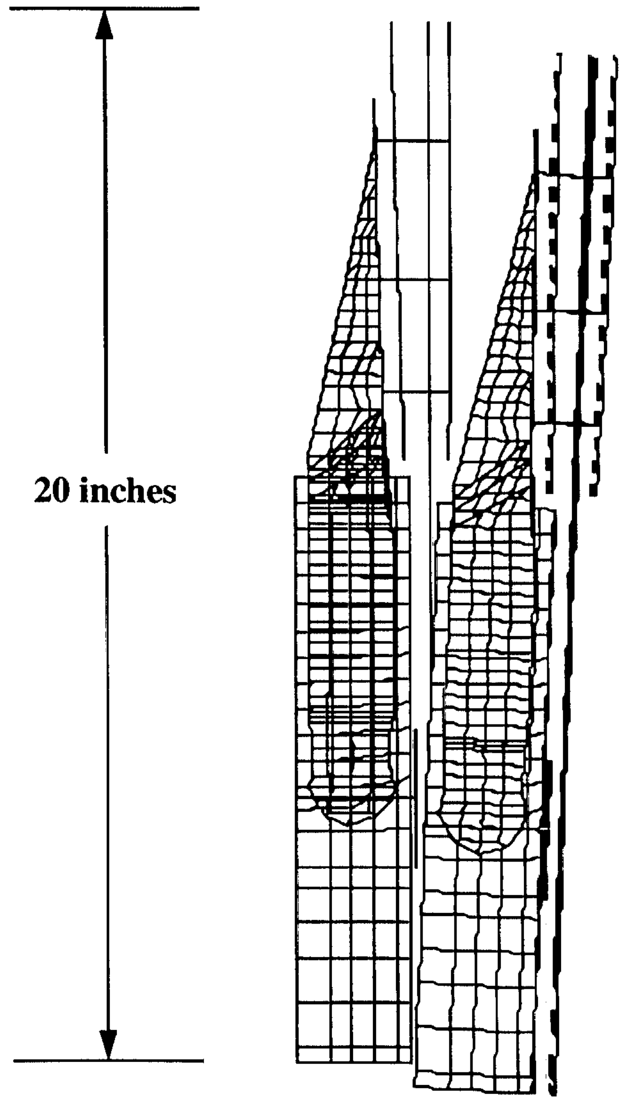
(b) Splice location 2.

Figure 4. Continued.



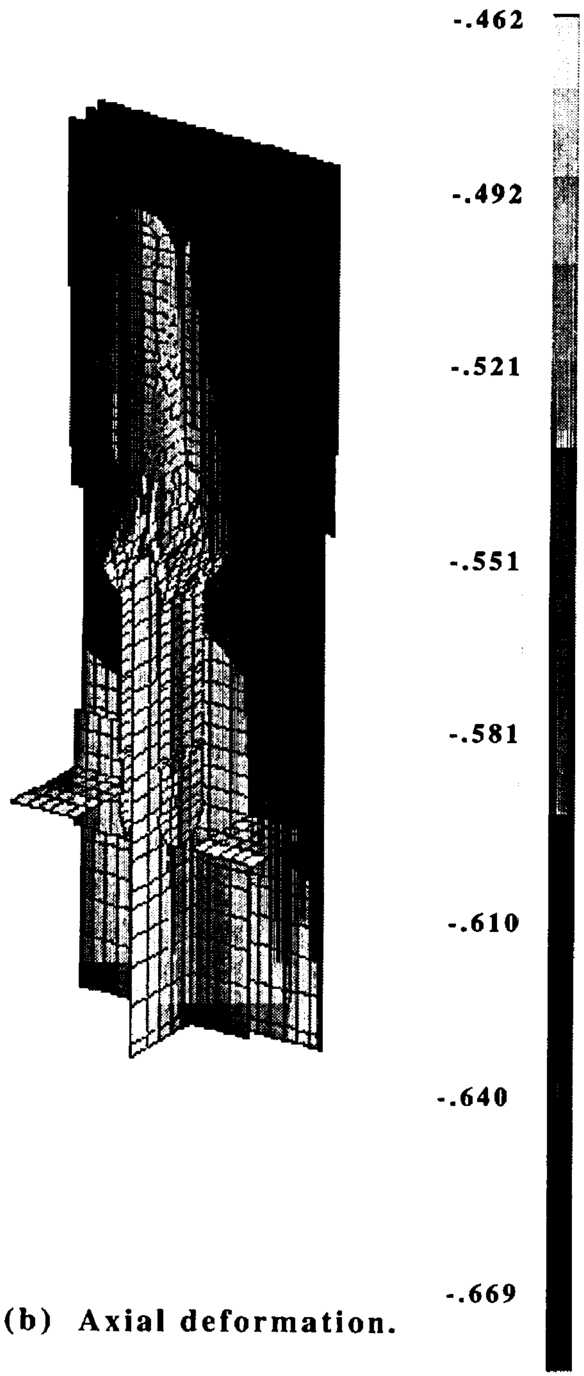
(c) Splice location 3.

Figure 4. Concluded.

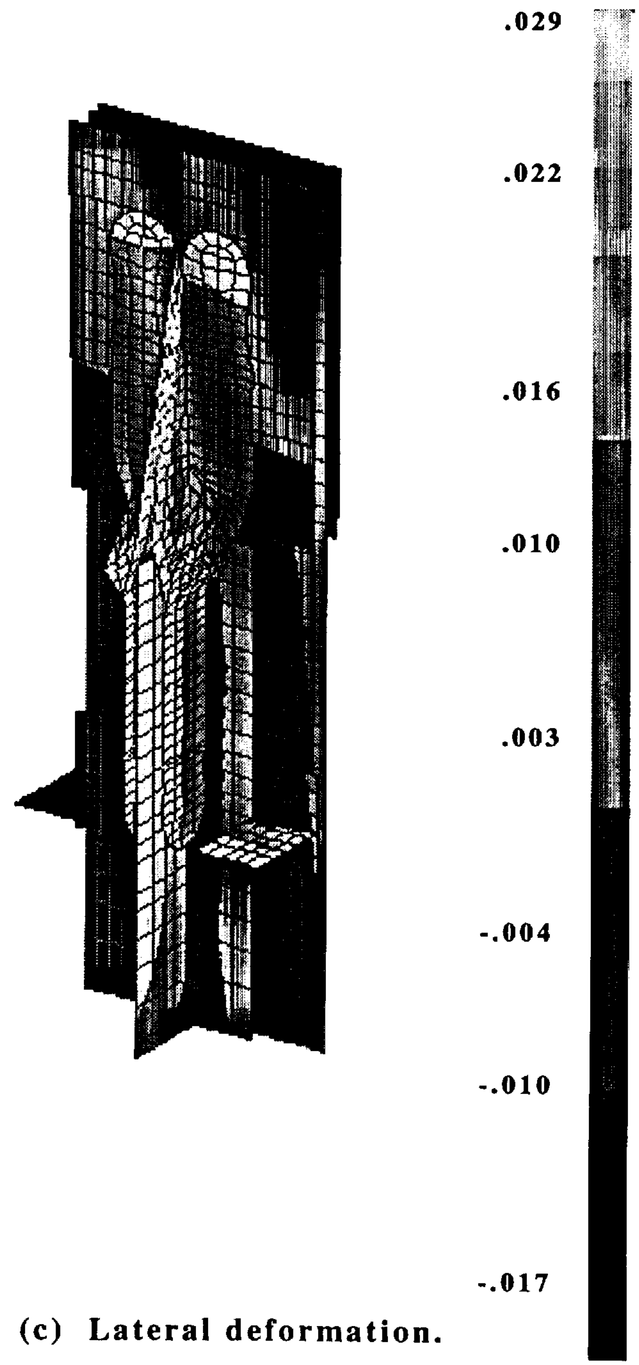


(a) Total deformation (deformation and model are to the same scale).

Figure 5. Deformation at Design Ultimate Load at splice location 1 (deformations are in inches).

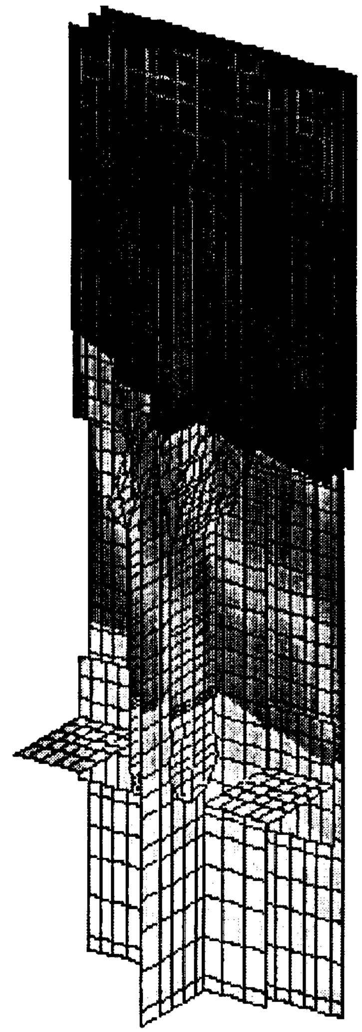


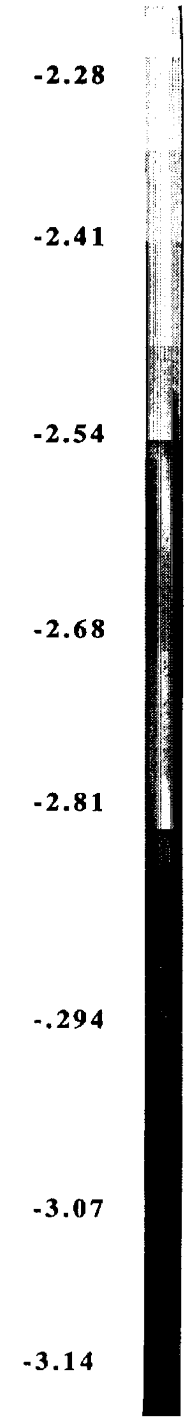
(b) Axial deformation.



(c) Lateral deformation.

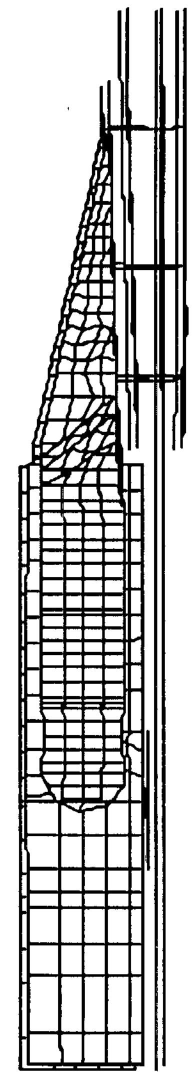
Figure 5. Continued.





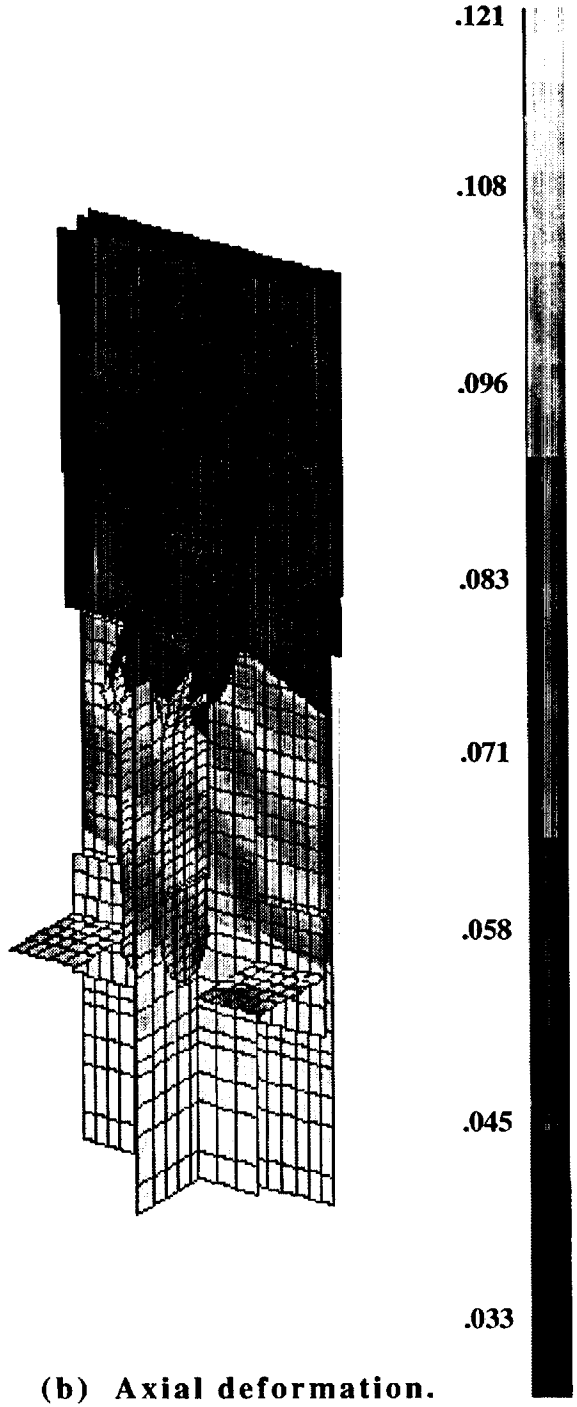
(d) Out-of-plane deformation.

Fgure 5. Concluded.

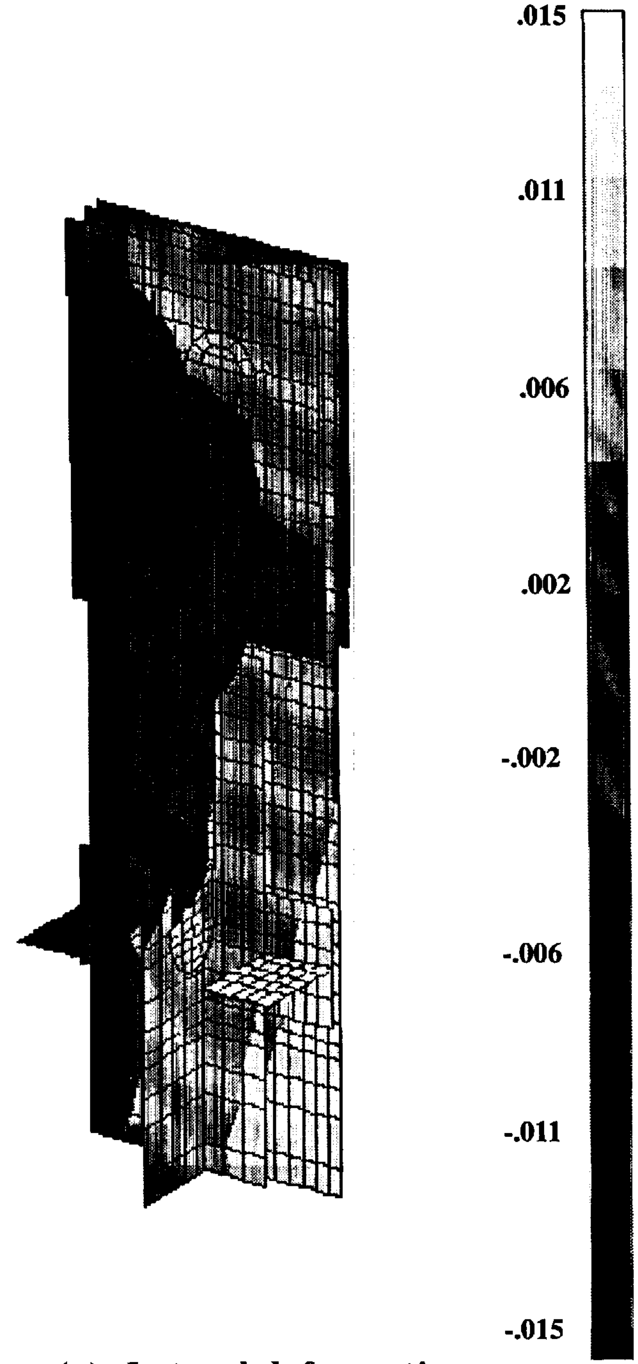


(a) Total deformation (deformation and model are to the same scale).

Figure 6. Deformation at Design Ultimate Load at splice location 2 (deformations are in inches).

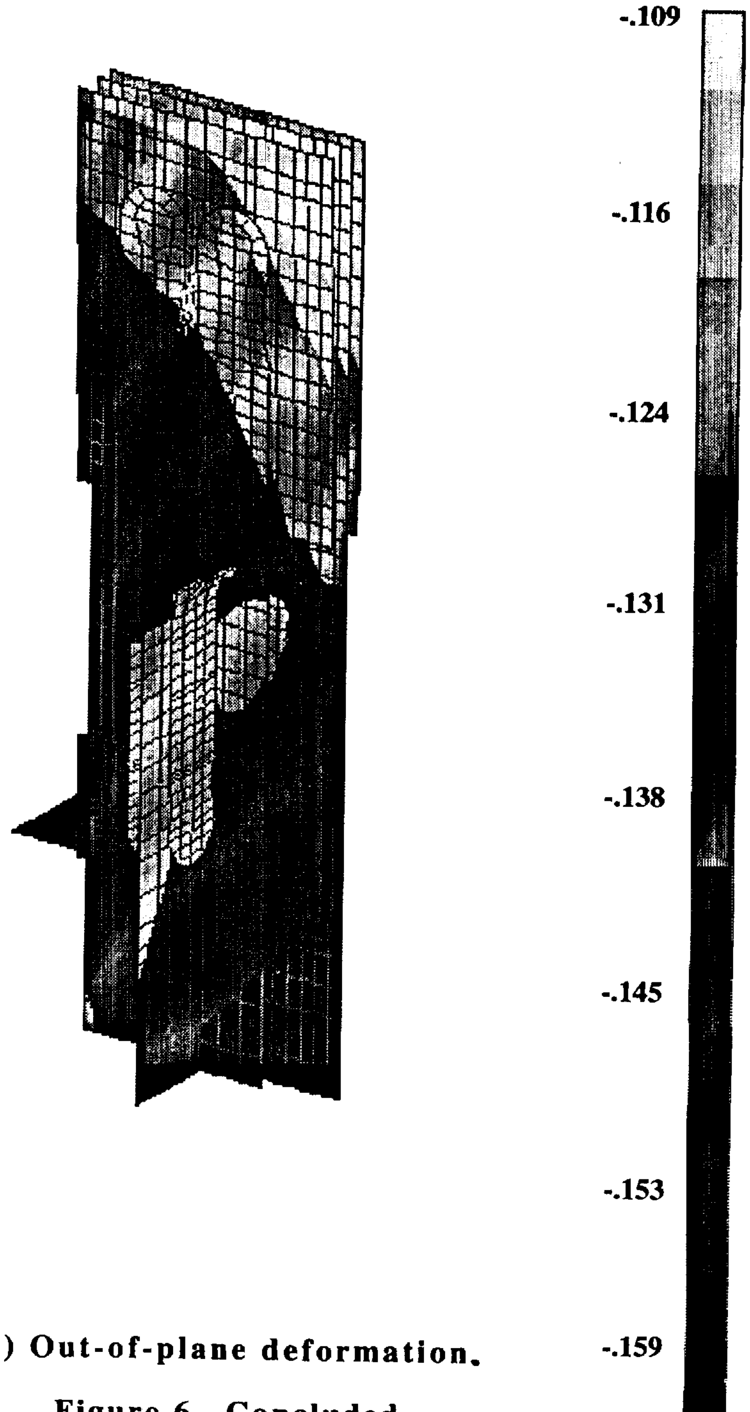


(b) Axial deformation.



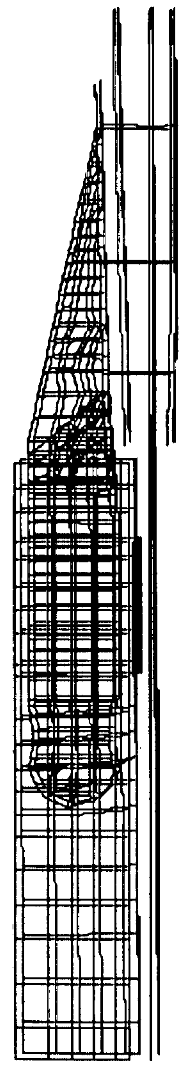
(c) Lateral deformation.

Figure 6. Continued.



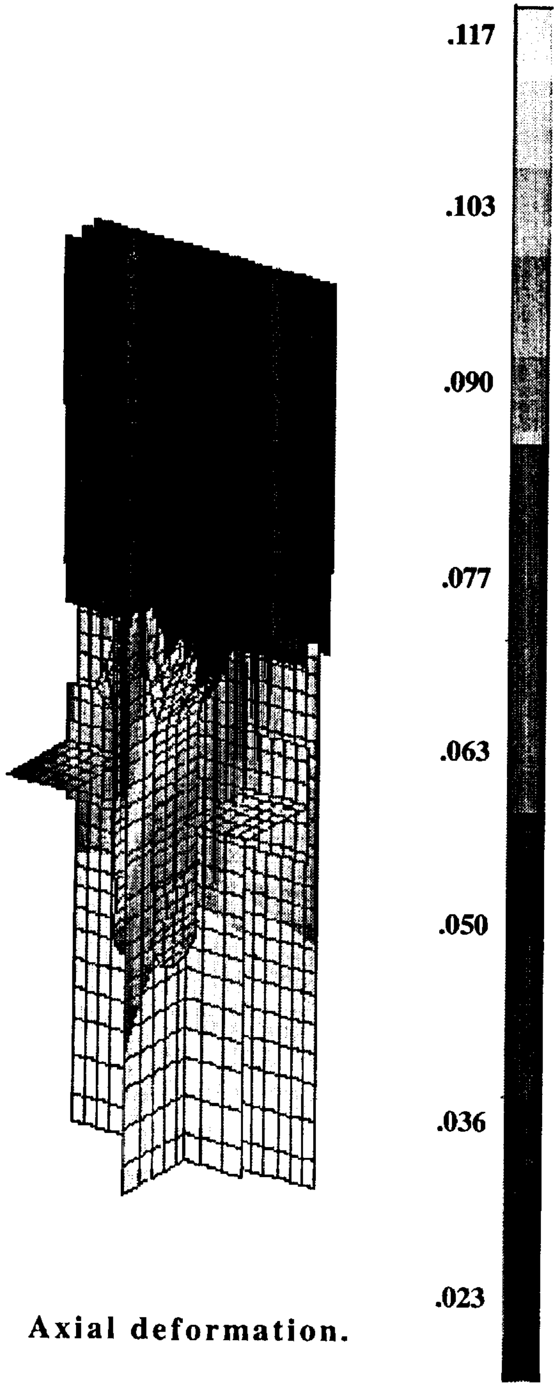
(d) Out-of-plane deformation.

Figure 6. Concluded.

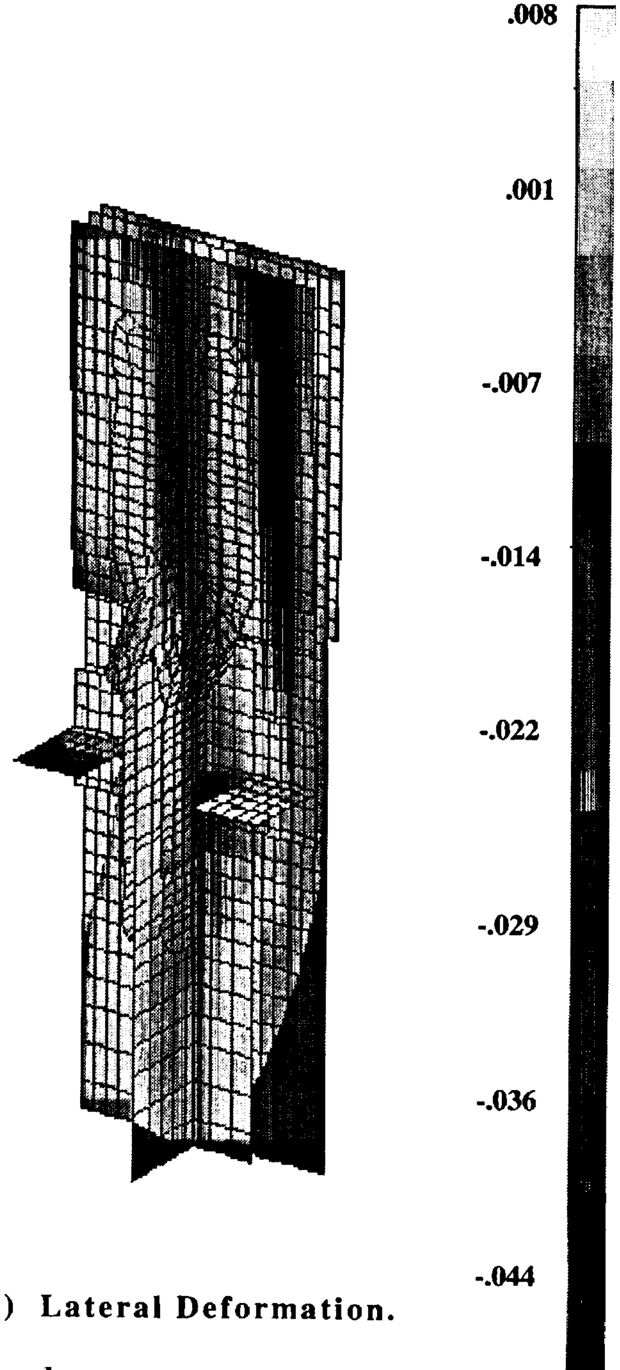


(a) Total deformation (deformation and model are to the same scale).

Figure 7. Deformation at Design Ultimate Load at splice location 3 (deformations are in inches).

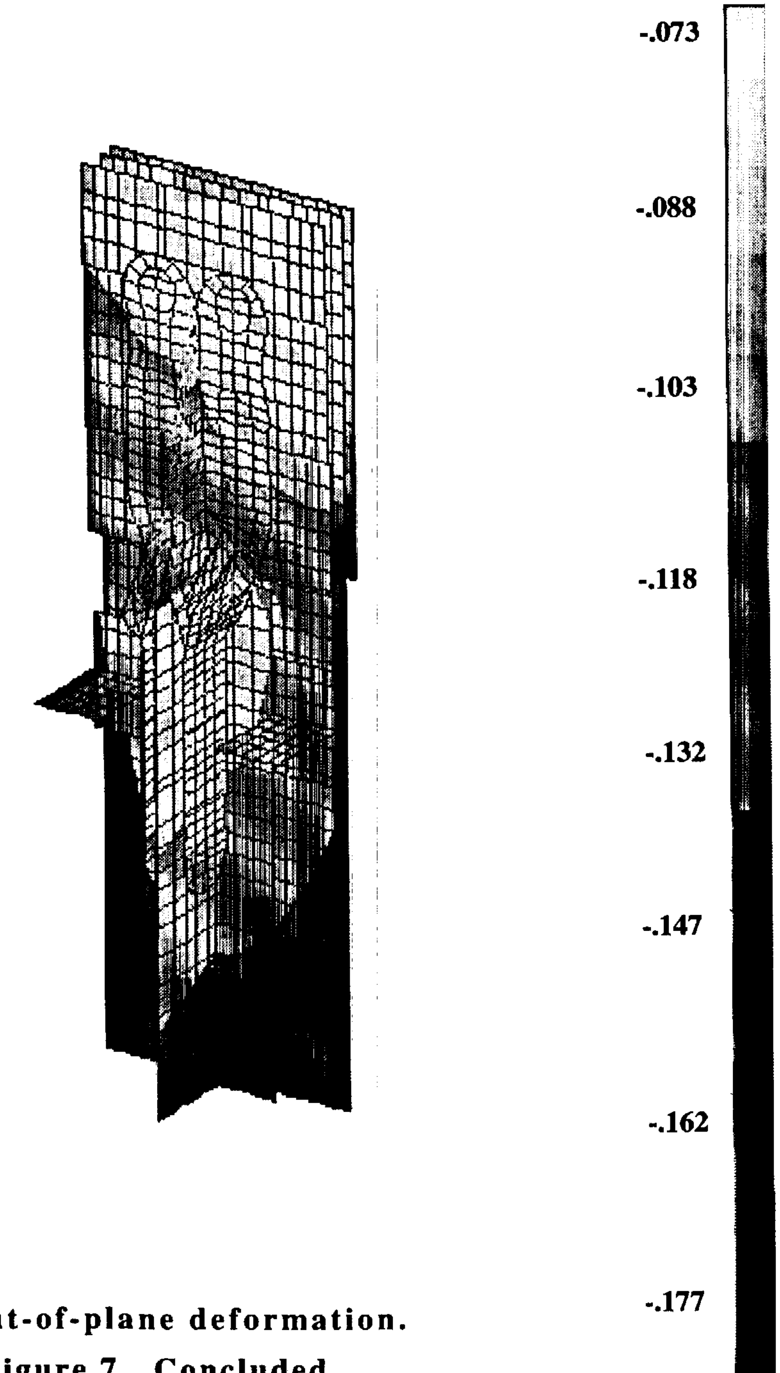


(b) Axial deformation.



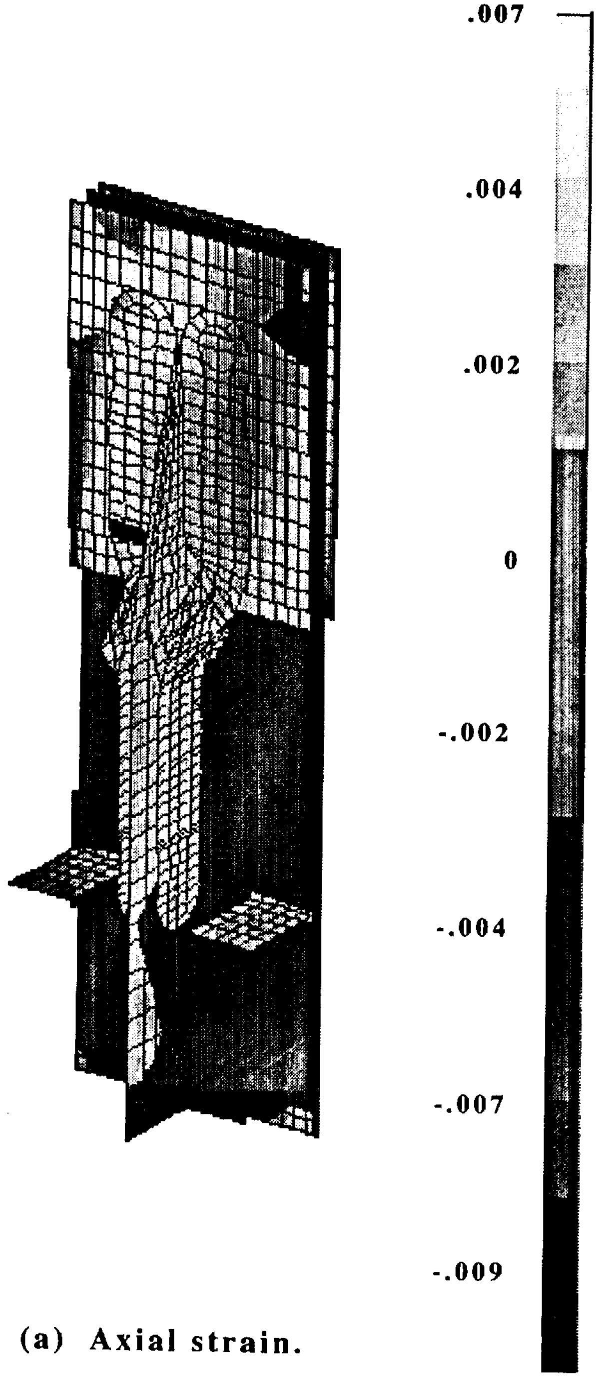
(c) Lateral Deformation.

Figure 7. Continued.

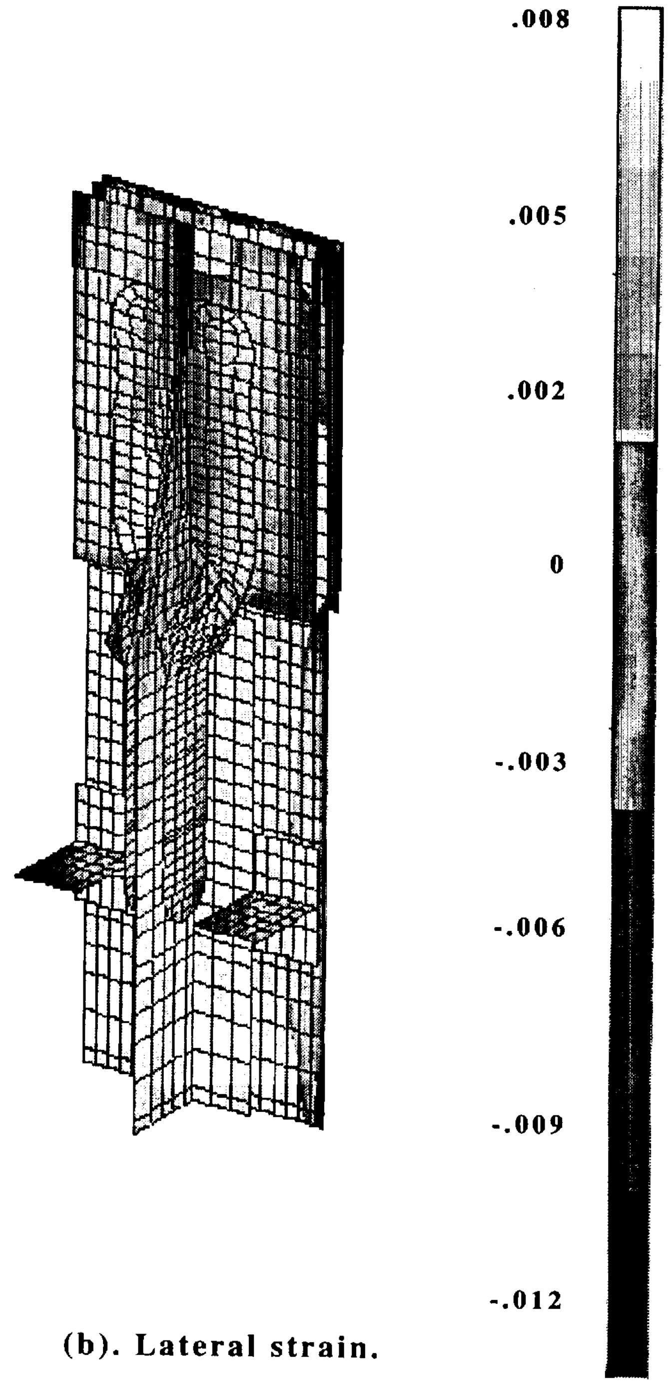


(d) Out-of-plane deformation.

Figure 7. Concluded.

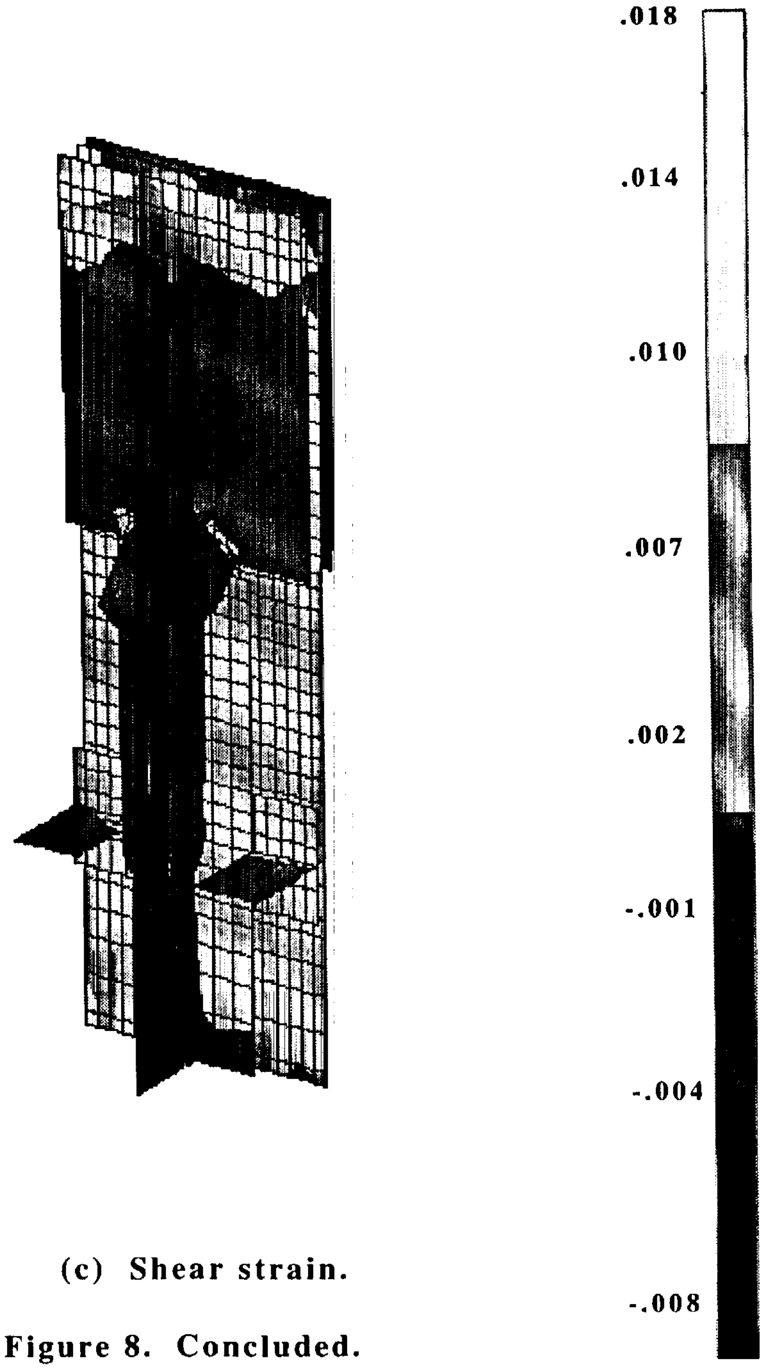


(a) Axial strain.



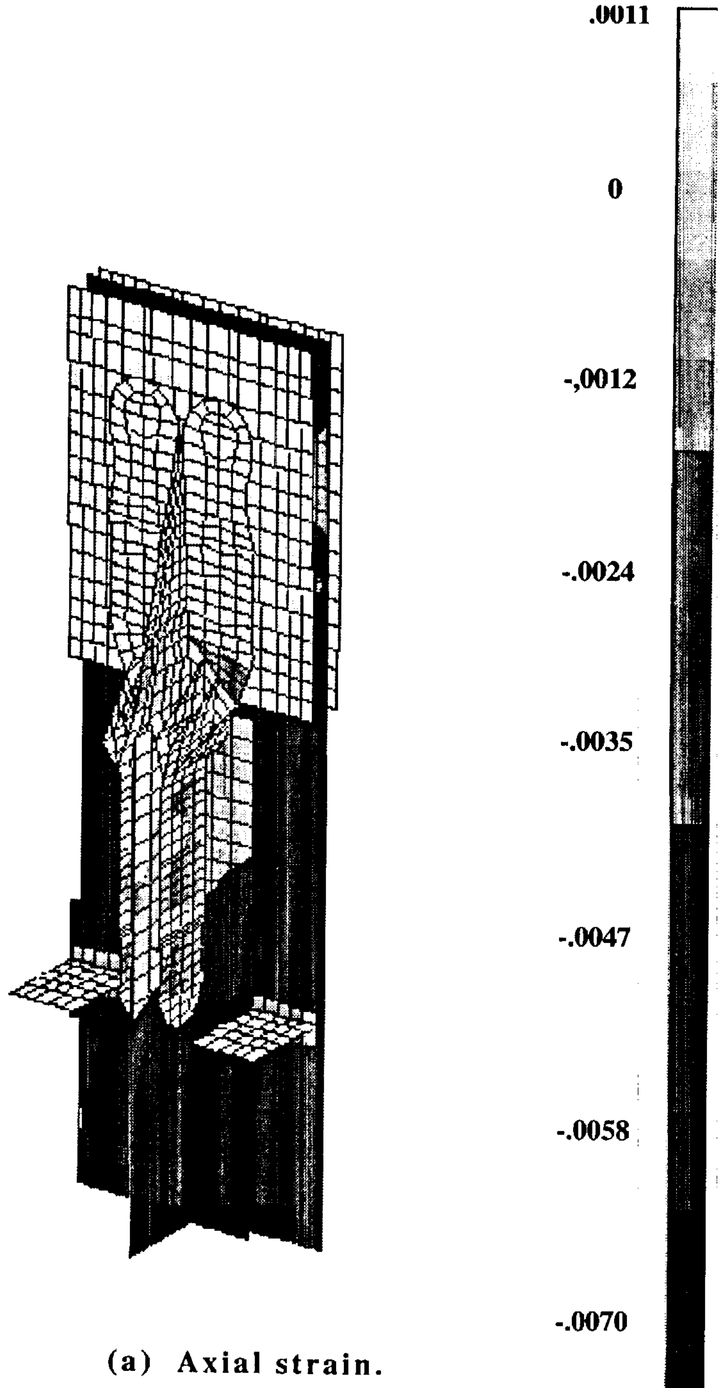
(b). Lateral strain.

Figure 8. Strains at Design Ultimate Load at splice location 1.

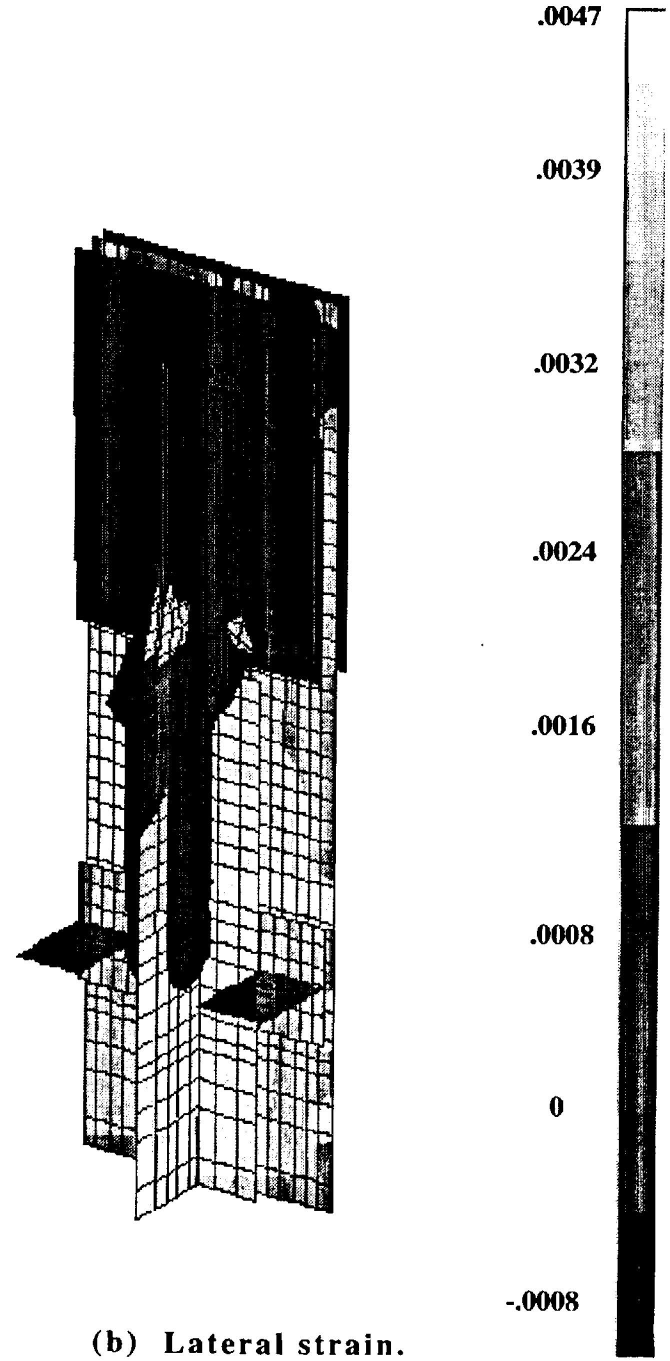


(c) Shear strain.

Figure 8. Concluded.

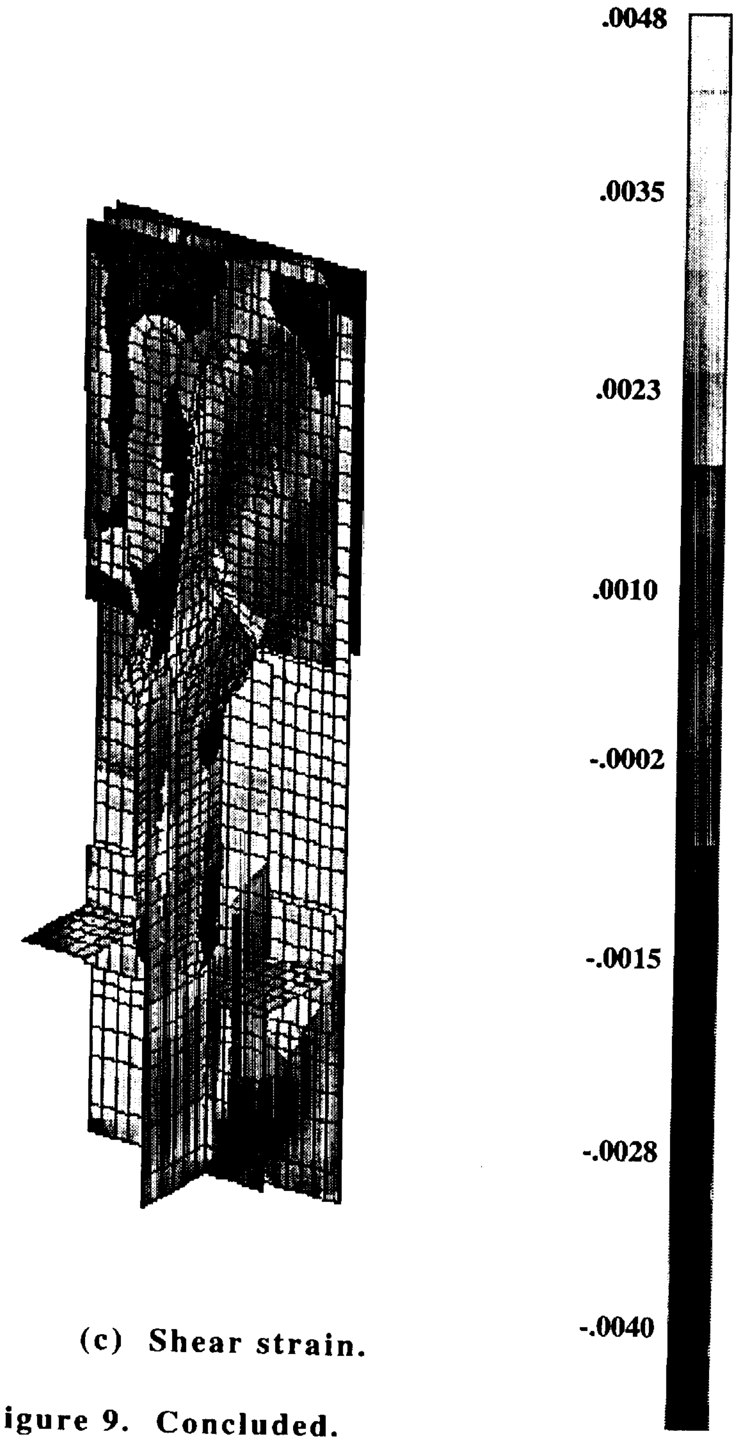


(a) Axial strain.



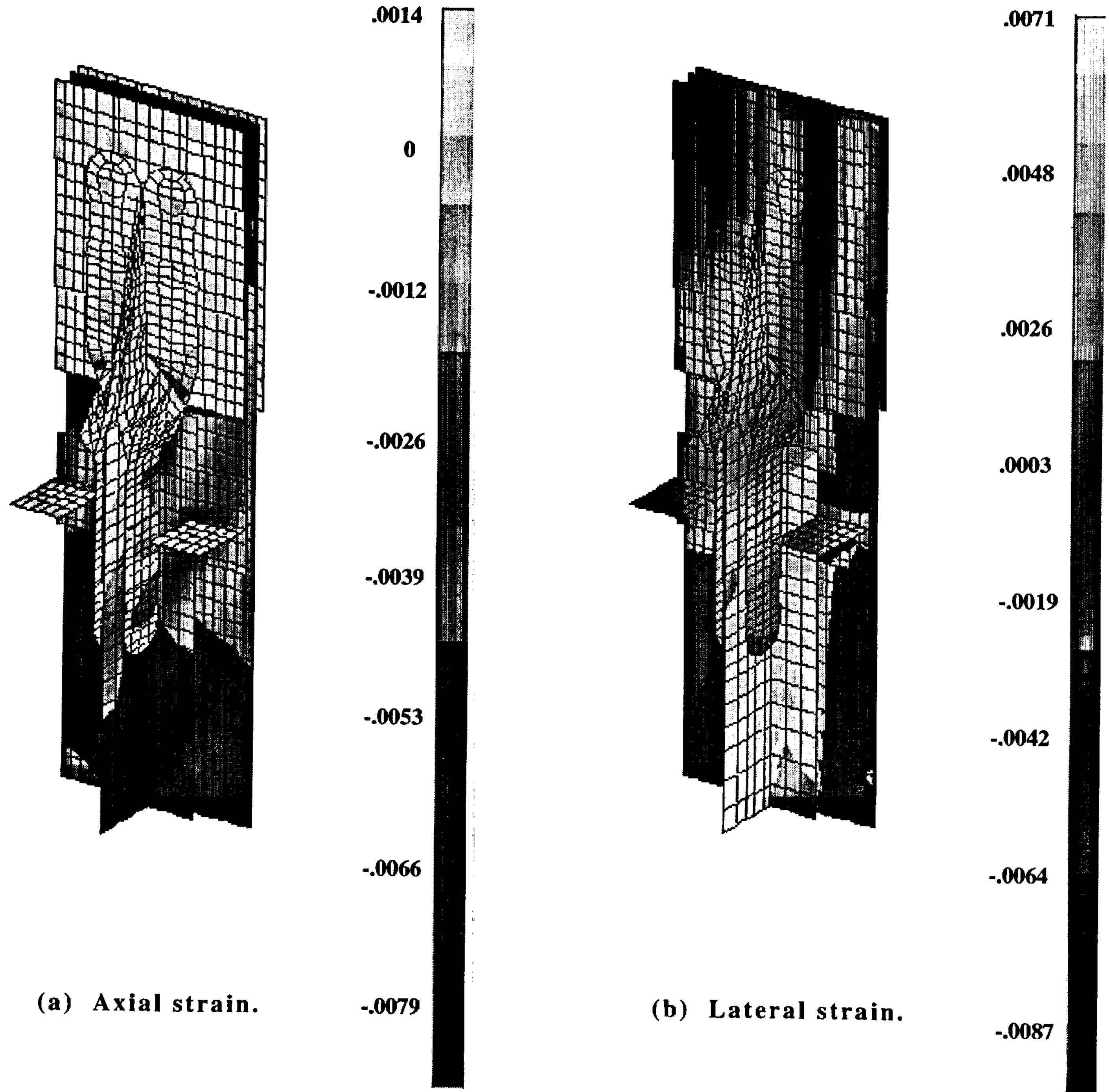
(b) Lateral strain.

Figure 9. Strain at Design Ultimate Load at splice location 2.



(c) Shear strain.

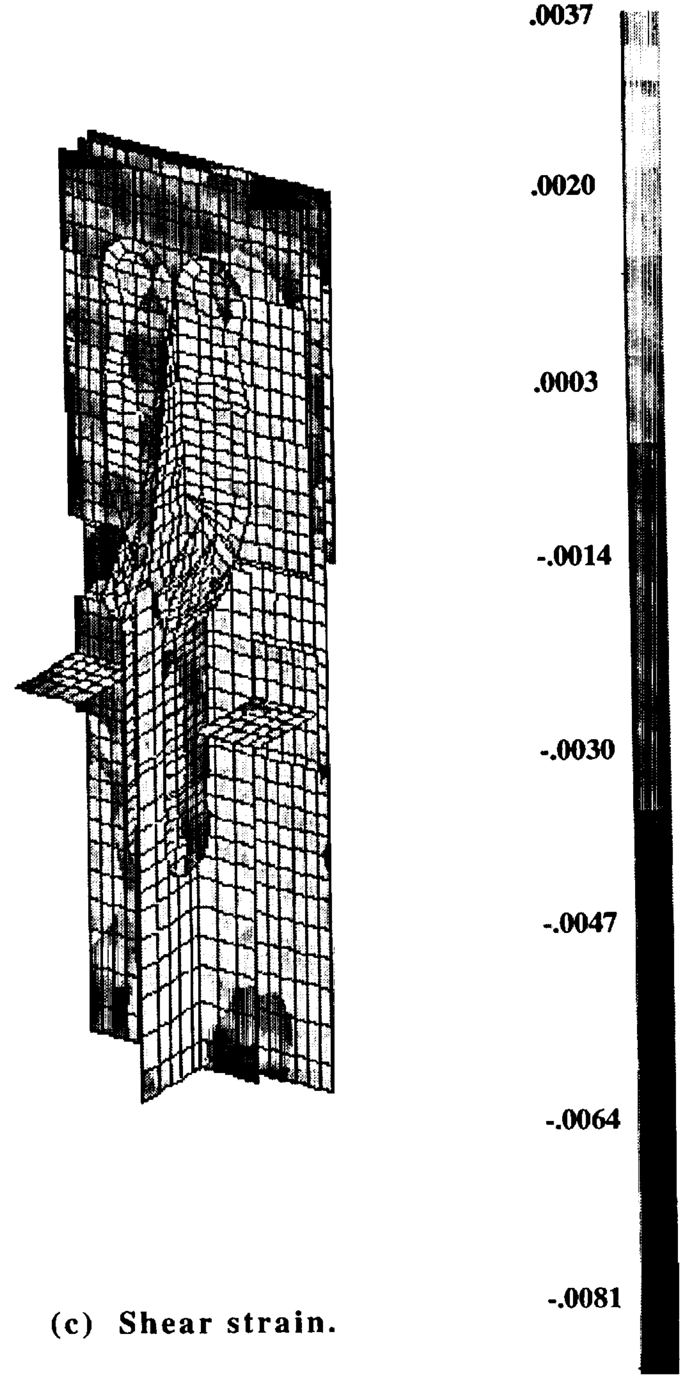
Figure 9. Concluded.



(a) Axial strain.

(b) Lateral strain.

Figure 10. Strains at Design Ultimate Load at splice location 3.



(c) Shear strain.

Figure 10. Concluded.

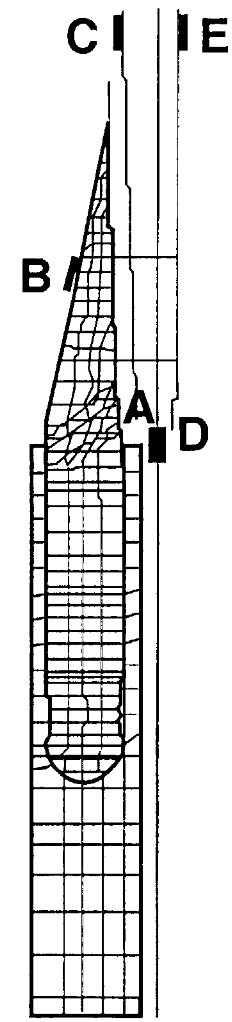


Figure 11. Strain gage locations.

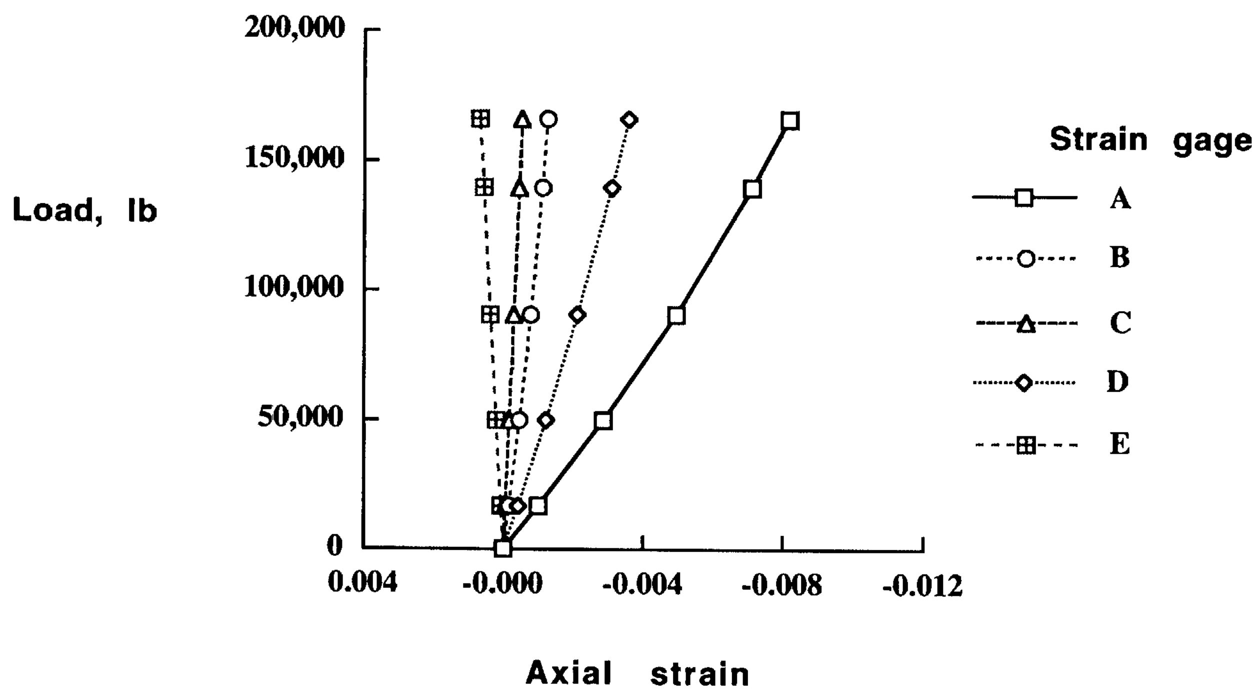


Figure 12. Axial strain predictions for the strain gages A through E at splice location 1.

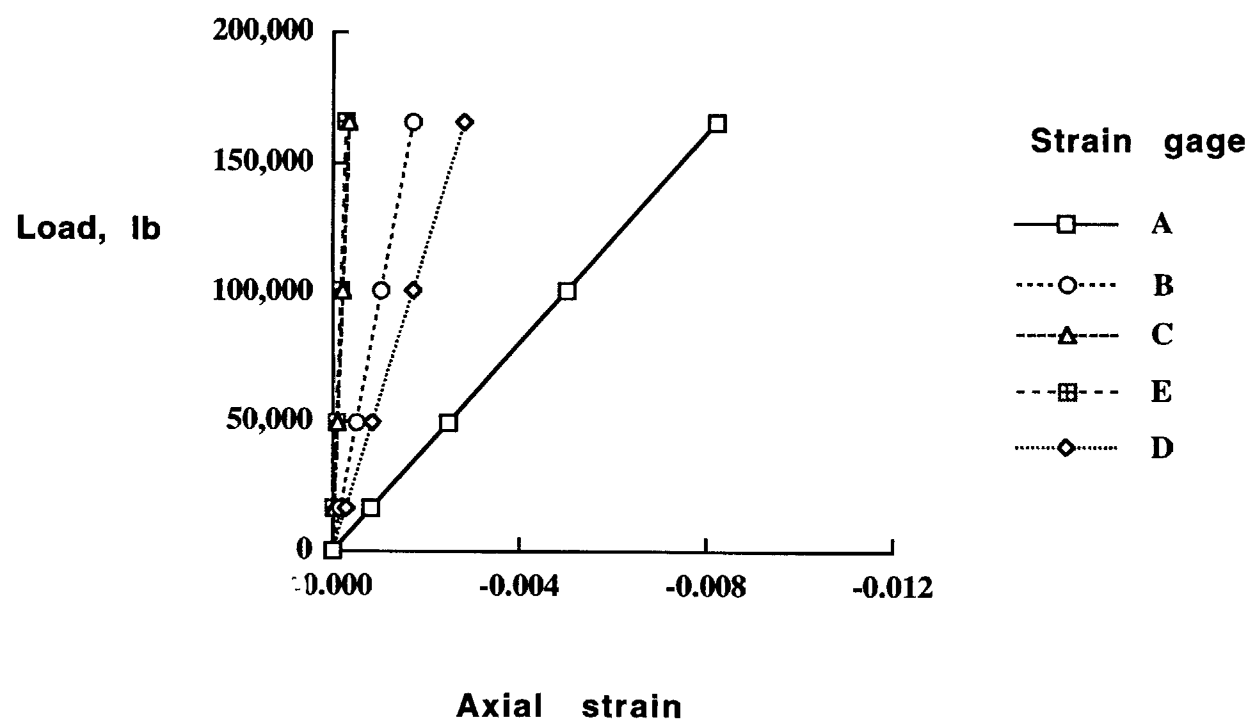


Figure 13. Axial strain predictions for strain gages A through E at splice location 2.

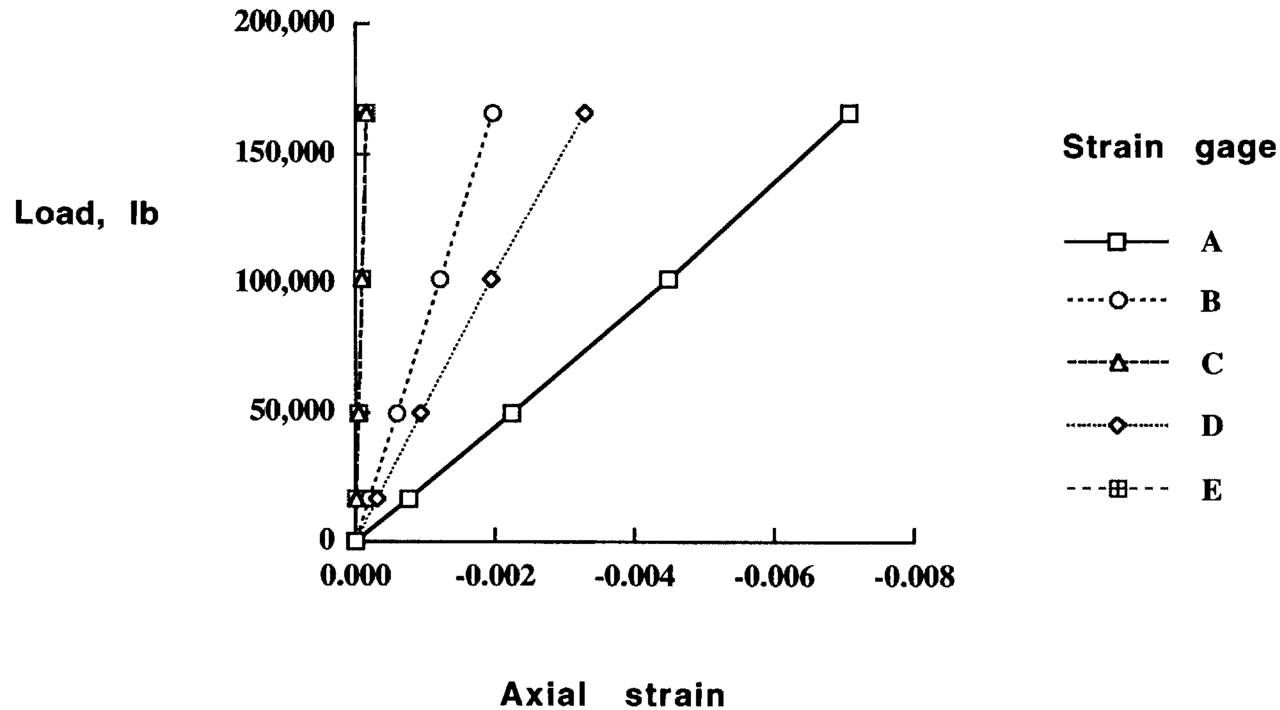


Figure 14. Axial strain predictions for strain gages A through E at splice location 3.

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
October 1995

3. REPORT TYPE AND DATES COVERED
Technical Memorandum

4. TITLE AND SUBTITLE
Analysis of Selected Compression Splice Joint Locations in a Graphite-Epoxy Transport Wing Stub Box

5. FUNDING NUMBERS
WU 510-02-12-01

6. AUTHOR(S)
Dawn C. Jegley

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
NASA Langley Research Center
Hampton, VA 23681-0001

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING / MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING / MONITORING AGENCY REPORT NUMBER
NASA TM 110170

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION / AVAILABILITY STATEMENT
Unclassified-Unlimited
Subject Category 39

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

Three critical compression splice joint locations in a stitched graphite-epoxy transport wing stub box have been analyzed to determined their expected structural performance. The wing box is representative of a section of a commercial transport wing box and was designed and constructed by McDonnell Douglas Aerospace Company as part of the NASA Advanced Composites Technology (ACT) program. The results of the finite element analyses of the splice joints are presented. The analysis results indicate that failure will not occur in the splice joint regions for loads less than the Design Ultimate Load of the wing box.

14. SUBJECT TERMS
composite material
bolted joint
stitching
finite element analysis

15. NUMBER OF PAGES
33

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	

NSN 7540-01-280-5500

Standard Form 298 (Rev. 2-89)
Prescribed by ANSI Std. Z39-18
298-102